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PROJECT OF A HIGH-ENERGY STANDARD PROPULSION UNIT FOR THE ELDO-B LAUNCH VEHICLE

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The ELDO (European Launcher Development Organization) development project for a 6-Mp hydrogen/oxygen rocket engine, to be used as standard propulsion unit for the high-energy upper stages of the series of ELDO-B launch vehicles, is described, with schematic sketches of single-engine arrangement in the third stage and four-engine clustering in the second stage of the three-stage launch vehicle. The specific impulses in vacuum (438.5 kp/sec/kg) and on liftoff, with preferential use of the pyrotechnic ignition method, are plotted. The pyrotechnic method offers the advantage of accelerated start of the turbopumps by the combustion gases of the powder trains and of elimination of an additional hydrogen pressurized tank. The thermodynamic principles, design data with two turbopumps, and future development prospects are given. A future project of a third (ELDO-C) three-stage variant, with a liftoff mass of 210 tons and a cluster of six engines in the second stage is described briefly, with plotted values of the thrust runup in the second and third stages.

Author

I. INTRODUCTION

At the beginning of 1965, the ELDO (European Launcher Development Organiza-

* Numbers in the margin indicate pagination in the original foreign text.

tion) sponsored a study on the development of a 6-Mp H_2/O_2 rocket engine in each of the following countries: England, France, and Federal Republic of Germany; this engine was to be used as a standard engine for the high-energy upper stages of the ELDO-B launch vehicles. These studies were let out on contract by the ELDO after a number of investigations (Bibl.1 - 4, 7) had definitized the two-stage launch system ELDO-B1 with a high-energy upper stage and the three-stage launch system ELDO-B2 with two high-energy upper stages, based on the Blue Streak as the first stage.

In the investigations on the launch system itself, the payload of a future three-stage ELDO-B2 launch vehicle was taken into consideration as a function of the thrust level of the second and third stage for various propellant masses in the second stage. Figure 1 shows the results of the investigations made in the pre-project stage, for the two missions of escape velocity and orbital velocity in 200km height (Bibl.3), in which case a constant specific vacuum impulse of 425 kp/sec/kg was assumed for the high-energy engines. Here, the liftoff mass was 110 tons, the thrust of the first stage was 144 Mp which corresponds to a liftoff acceleration of 0.3 G, and the maximum mass of the upper stages and payload was 23 tons. It was found that, at optimum layout of the upper stages, the thrust level of the different stages could be varied within certain limits, without greatly reducing the payload.

This offers the possibility of developing a single high-energy power plant which, at a thrust of about 6 Mp, could be used as the only engine for the third stage or, arranged as a cluster of four engines with a total thrust of 24 Mp, in the second stage of the three-stage launch vehicle. This high-power standard propulsion unit actually meets the deadline and cost estimates of the ELDO. /4

The arrangement of four engines in the second stage is shown in Fig.2.

Figures 3 and 4 give a schematic view of the two launch vehicles ELDO-B1 and ELDO-B2 which are used as basis for our discussions. Both launch vehicles have the same top stage. The B2 launch vehicle, in addition, has a modified first stage. The values in brackets refer to the specific impulses which, according to detailed engine tests, actually appear realizable.

It can be expected that the standard propulsion unit, to be developed within the framework of the ELDO-B program, will become the high-energy standard power plant of the ELDO for the next 10 - 15 years. For this reason, the engine, by proper layout, must be given adequate adaptability to satisfy the presently predictable possibilities of future ELDO missions. In this connection, the three-stage launch vehicle, discussed last year as a possible ELDO-C version and having a liftoff mass of 210 tons should be mentioned (Bibl.6, 7). In a second stage, this variant was to carry six high-energy propulsion units. Figure 5 shows the resultant thrust runup in the second and third stage (Bibl.6).

II. DESIGN AND LAYOUT OF THE PROPULSION UNIT /5

1. Description of the Engine

A schematic sketch of the high-energy propulsion system, using one engine in the smaller upper stage, is shown in Fig.6. As a typical example, the propellant feeding is done here by two turbopumps. The oxygen and hydrogen are carried in the propellant tanks under a certain overpressure which is produced by gaseous helium. A second possibility is pressure gas generation in the hydrogen tank by gaseous hydrogen, heated in the cooling jacket of the combustion chamber. This positive pressure prevents cavitation at the pump inlet during propellant feed. The two pumps deliver about 97 - 98% of the propellant to the combustion chamber and the remainder to the gas producer. Here, the combustion gases are produced that supply the turbines used as pump drive. After a partial expansion in the turbines, an afterexpansion of the combustion gases takes place either in the large jet nozzle or, as shown in Fig.6, in a separate exhaust nozzle, thus producing a slight auxiliary thrust of the order of magnitude of 120 kilopond.

In contrast to the medium-energy propellants used in the upper stages of the ELDO-A launch vehicle, hydrogen and oxygen are not a hypergolic fuel. Therefore, in any H_2/O_2 engine, an ignition system must be installed for igniting the propellant in the gas producer and in the combustion chamber. Of the three known ignition methods, namely, electric, pyrotechnic, and hypergolic methods, the two former are directly in question, with the pyrotechnic method presumably being the preferred process (Fig.6). In the case of a limited number of re-ignitions, this method has the advantage of simplicity and great reliability as well as of the possibility to accelerate the start-up of the turbo- /6 pumps by the combustion gases of the solid propellant charge. This also eliminates the necessity of an additional H_2 pressurized tank, which is needed in the electric ignition method to obtain the desired short start-up times for reaching full thrust.

The combustion chamber and the bell-shaped jet nozzle are designed in tubular form and are provided with regenerative cooling, using the hydrogen as coolant before its introduction into the combustion chamber (Fig.7). The thin-walled small tubes, forming the jacket, have a double-cone shape and are made of tempered steel. For cooling technology and design reasons the number of tubes near the nozzle end is twice as great as in the combustion chamber itself. The liquid hydrogen which has a temperature of about 25° K, enters the cooling jacket at a certain distance below the narrowest cross section and flows first into the tubes completing the nozzle, downward toward the end of the nozzle. A reverse manifold is used for distribution over the remaining tubes and for reflux to the injection head where the hydrogen finally enters the combustion chamber in the gaseous state, through a large number of orifices. The oxygen, in the liquid state, is injected through separate orifices which individually are surrounded by a group of hydrogen orifices. The regeneratively cooled jet nozzle, whose exit cross section is determined by the power of the diffuser used in the test operation, can be extended by a radiation-cooled section.

2. The Turbopump System

For the high-energy standard engine, three different turbopump systems are in question. In principle, these differ only by whether the two fuel pumps /7 are driven by two turbines, as shown in Fig.6, or by a single turbine. If only one turbine is used, the two pumps can be mounted to a single shaft and rotate at the same rpm (Fig.8) or else can operate at different rpm by the use of an auxiliary gearing (Fig.9).

The three turbopump systems can best be compared if the same specific performance is used as basis for both pumps and turbines and if the system-specific engine impulse is considered, which contains the differing structural weight. Because of the elimination of an additional structural component (gear unit or second turbine), the single-shaft arrangement shown in Fig.8 represents the simplest design but gives the lowest system-specific impulse. On the other hand, the system-specific impulse is greatest in the gear design shown in Fig.9. However, here the gearing must be designed as an auxiliary component which, because of the high temperature gradient and the lubrication by the hydrogen, raises quite some special development problems. In the two-shaft design, as shown in Fig.6, a possibility exists to use the same single-stage turbine, at differing admission, for driving the hydrogen as well as the oxygen pump; this will keep the additional development expenditure, required by the second turbine, within moderate limits.

In all three turbopump systems, one O_2 and one H_2 control valve, installed in front of the gas producer, will be sufficient for controlling the rpm of the turbopumps in stationary operation, which can be used also for adjusting the thrust level. Two additional control valves, installed into the H_2 and O_2 lines leading to the combustion chamber, can be used for regulating the mixture ratio of the combustion chamber as well as the thrust of the engine.

III. THERMODYNAMIC LAYOUT OF THE PROPULSION UNIT /8

1. Thermodynamic Principles

The high-energy propulsion unit, to be developed for the ELDO, is somewhat similar to a well-known American engine so far as thrust, mixture ratio, and expansion ratio are concerned. Therefore, it can be assumed that, after the development of the ELDO propulsion unit is concluded, the engine will at least have the same values of combustion efficiency, friction coefficient, and recombination in the jet nozzle. A check calculation of the individual nozzle losses, measured on the American engine, showed that it is presently possible to precalculate nozzle efficiencies of H_2/O_2 upper stage engines with considerable accuracy. This is clearly expressed in today's attitude of the US industry, which restricts experiments of this type to a minimum in new designs. Therefore, it seems advisable to use the specific performance factors, experimentally determined in USA, as a basis in designing the ELDO engine and in determining its specific vacuum impulse.

The altitude simulation of the combustion chamber and of the engine, with respect to output and power, presupposes a full-flow nozzle. The most economical method of such altitude simulation is the use of a supersonic diffuser installed aft of the jet nozzle and compressing the exhaust gases from the nozzle - end pressure to atmospheric counterpressure, by means of the kinetic exhaust-gas energy (Fig.18). The diffuser power, characterized by the pressure ratio within the diffuser, is predominantly a function of the combustion-chamber pressure and the mixture ratio. This power determines the maximum area ratio of the regeneratively cooled jet nozzle. The diffuser power, and thus the area ratio, of the nozzle are greater the higher the combustion-chamber pressure.

In Fig.10, the maximum area ratio and the maximum expansion ratio of the jet nozzle are plotted as a function of the thrust and of the combustion- /9
chamber pressure, for various mixture ratios in the combustion chamber. During this process, the exhaust gases are compressed to the ambient pressure by an optimal supersonic diffuser. According to Fig.10, for a 6-Mp engine whose combustion-pressure chamber is 36 kp/cm^2, the maximum area ratio of the nozzle, limited by the diffuser power, will be $A_e/A_t = 60$*. If the same engine is to be operated on the test stand with a thrust of 5 Mp and thus with a lower combustion-chamber pressure, the maximum area ratio will decrease to $A_e/A_t = 50$.

As shown by American examples, a suitable design of the upper stage engine makes it possible to omit a part of the development tests at atmospheric counter-pressure conditions and thus to do away with an altitude simulation. This has to do mainly with the behavior of the combustion chamber in the case of separa-

* The mixture ratio in the combustion chamber will be denoted by m.

tion of flow. As a typical example, Fig.11 gives the slope of the wall temperature along a 6-Mp H_2/O_2 tubular combustion chamber, whose thrust nozzle has an area ratio of 50. In the first case, full nozzle flow is maintained by a diffuser connected in series. In the second case, the flow separates on the indicated point, at atmospheric counterpressure. However, the generation of impermissibly high temperatures is avoided.

2. Combustion-Chamber Pressure, Expansion Ratio, and Mixture Ratio

In Fig.12, the effective specific vacuum impulse of the engine is plotted as a function of the combustion-chamber pressure, for a 6-Mp engine with a mixture ratio of 5 kg O_2/kg H_2. The diagram indicates the specific vacuum impulse for the following two cases:

a) The maximum nozzle-end diameter D_e, on the basis of installation restrictions in the second stage of the ELDO-B2 launch vehicle, is given.

b) The nozzle area ratio is determined by the critical output of the supersonic diffuser used in experimental operation.

In both cases, the calculation is made with and without utilization of /10 the turbine exhaust gases for thrust production. The distinct maxima are explained by the increase in friction losses, due to the enlargement of the nozzle area ratio, and by the increase in throughput through the gas producer, due to the rise in combustion-chamber pressure.

According to Fig.12, the optimum combustion-chamber pressure is between 36 kp/cm^2 and 40 kp/cm^2, if the turbine exhaust gases are used for thrust production. Depending on the combustion-chamber pressure, the gain in specific vacuum impulse of about 4 - 7 kp/sec/kg, produced by an enlargement of the area ratio to 88, cannot be tested in a simple diffuser system. For comparison, the specific impulse of an American engine is plotted, which has a combustion-chamber pressure of 21.2 kp/cm^2, a nozzle area ratio of 40, and a turbopump capacity based on the main-flow system.

Figure 13 shows the effective specific vacuum impulse of both combustion chamber and power unit as a function of the nozzle area ratio, for a combustion-chamber pressure of 36 kp/cm^2. Because of the installation restriction in the larger upper stage, the maximum area ratio obtainable is 88, which is below the optimum ratio even if the structural weights of jet nozzle and transition structure are taken into consideration.

This yields the following area ratios for the jet nozzle and the corresponding effective specific vacuum impulses of the propulsion unit (p_c = = 36 kp/cm^2, m = 5 kg O_2/kg H_2).

a) Propulsion unit with fully regenerative cooling and a minimum thrust of S = 5 Mp, respectively, S = 6 Mp:

A_e/A_t = 50 and I_{sp}^{vac} = 428 kp/sec/kg,

resp.

A_e/A_t = 60 and I_{sp}^{vac} = 432 kp/sec/kg;

b) Propulsion unit with flanged radiation-cooled jet nozzle /11

A_e/A_t = 88 and I_{sp}^{vac} = 439 kp/sec/kg.

The influence of the mixture ratio on the effective specific vacuum impulse of combustion chamber and power plant is plotted in Fig.14. The diagram is again based on the maximum nozzle-end diameter, given by the installation restriction in the second stage. Accordingly, the optimum mixture ratio in the combustion chamber, correlated with the maximum specific engine impulse, is located near 4.5 kg O_2/kg H_2. To obtain a statement on the optimum mixture ratio, coordinated with the maximum payload, the system-specific impulse I_q is also plotted in Fig.14, but only taking the variations in structural weight determined by the mixture ratio into consideration. The qualitative slope of this system-specific impulse is sufficient for defining the optimum mixture ratio. It is found that the mixture ratio of the engine, optimum with respect to the payload, is of the order of 5 kg O_2/kg H_2, at which value also the American H_2/O_2 engines are operated.

3. General Design Viewpoints of the Turbopumps

In the USA, the viewpoints decisive for the design of pumps and turbines has been subjected to detailed overall optimizing considerations. Some of the discussed viewpoints are compiled in Table 1. This scheme can be used for designing the high-energy ELDO engine which, at first, is to operate at a moderate thrust of 6 Mp and later possibly in a higher thrust range. Taking into consideration great reliability, high specific impulse, and a reasonable structural expenditure, the following picture will result:

The oxygen pump is designed as a single stage. This is done because of /12 the high specific rpm and the resultant possible high pump efficiency. In addition, the oxygen pump, at the mixture ratio of 5 kg O_2/kg H_2 requires only about 20% of the total drive power, so that an increase in number of stages merely would result in a higher structural weight but would not produce a noticeable improvement in the specific engine impulse.

The conditions are different for the hydrogen pump which absorbs about 80% of the propulsive power. Here, the use of a two-stage pump will produce an improvement in efficiency by 10 units, a corresponding decrease in the gas-producer throughput, and thus a concomitant increase of the specific impulse. On the basis of similitude laws, the structural weight of a two-stage pump is not much greater than that of a single-stage pump. In addition, a two-stage pump (with respect to a later thrust augmentation) offers the possibility of more favorable peripheral speeds and blade angles, as indicated in Fig.15.

The optimum number of turbine stages can be estimated, for the turbopump systems in question, on the basis of Fig.16. In this diagram, the specific vacuum impulse of the engine is plotted as a function of the number of turbine stages. By a more accurate investigation, which takes the differing structural weights into consideration, it can be demonstrated that the optimum solution is one two-stage turbine which directly, or over a gear unit, drives both pumps, or else two single-stage turbines each of which is mounted to a common shaft with one pump.

4. Survey over the Preliminary Design Data

Figure 17, on the basis of a typical example, gives a general view over the project data of the 6-Mp H_2/O_2 engine, where the case of propellant feed /13 by means of two turbopumps is considered. All results, discussed above, are contained therein.

IV. DEVELOPMENT OF THE PROPULSION UNIT

For insertion into orbit as well as for obtaining attitude control and stage separation with the launch vehicle, the engines must satisfy certain specifications, within definite tolerances. This is the main goal of the engine development. Table 2 shows several important specifications, compiled from data obtained with US upper-stage engines and used as a basis, for the time being, for the high-energy engine to be developed for the ELDO.

To proceed with the development, which has the goal of meeting these specifications, a number of test stands are required. For financial reasons, however, their number will be much less than that of American test stands which were designed for the development of the two high-energy engines RL 10 and J 2. In both of these propulsion units, the Battleship test stands were supplemented by five engine test stands and about 20 component test stands. This equipment permitted the development of propulsion units over a period of three years up to PFRT (Preliminary Flight Rating Test) and within four years up to completion of the reliability tests.

Deviating from the absolute minimum of using only one test stand each for engine and components, the number of test stands listed in Table 3 would be required. During the development period, these test stands will be occupied by an extensive experimental program, as shown in the example given in Table 4. /14
This Table contains several development projects for the engine test stands. Figure 18 gives a schematic view of the test setup for the development of combustion chamber and jet nozzle. According to this example, the altitude simulation of ignition is done by a vacuum pump and a full-flow nozzle, using a diffuser. It is obvious that, in view of the multiplicity of development problems and the limited number of test stands, the development of the high-energy propulsion unit cannot be concluded within the scheduled time of 3 - 4 years. Therefore, a mean development time of 5 years is expected.

V. SUMMARY

The H_2/O_2 propulsion unit, to be developed for the ELDO, is designed primarily for driving the high-power upper stages of the ELDO launch vehicles Bl and B2. Despite its present design of realizing maximum specific impulse at moderate thrust, this presumable high-energy standard power unit of the ELDO for the next 10 - 15 years, does not exclude the possibility of a later development in the direction of higher thrusts and of a throttleable engine. In the preliminary design, a thrust of 6 Mp is obtained, at a combustion-chamber pressure of 36 - 40 kp/cm^2 and a mixture ratio of the order of magnitude of 5 kg O_2 per kg H_2. For this, an optimum specific vacuum impulse of 438.5 kp/sec/kg can be computed, using the specific power factors obtained in the USA with a similar engine. In the first development phase, the engine will be provided only with a regeneratively cooled jet nozzle, while the second development phase provides for a radiation-cooled part so as to ensure an optimum area ratio. The development period of the engine is estimated to be a minimum of four years up to the preliminary flight rating test.

TABLE 1

DESIGN VIEWPOINTS FOR TURBOPUMPS IN LAUNCH VEHICLES

1. Efficiency at the design point
2. Efficiency in the design range
3. Stability of the characteristic in the control range
4. Maximum peripheral speed and maximum obtainable pump pressures
5. Blade exit angle, under consideration of the stability and efficiency characteristics in the design range
6. Bearing
7. Lubrication
8. Compensation of axial thrust
9. Packing
10. Mass inertia moments and vibration behavior
11. Structural weight
12. Manufacture

TABLE 2

SPECIFICATIONS FOR DEVELOPMENT OF THE 6-Mp H_2/O_2 PROPULSION UNIT

$$\left.\begin{array}{l}\text{1. Vacuum thrust}\\ \text{2. Specific vacuum impulse}\\ \text{3. Mixture ratio}\end{array}\right\}\ \text{in stationary operation}$$

4. Time and total impulse before ignition up to 90% or 100% of rated thrust
5. Time and total impulse from combustion cutoff command up to (for example) 10% of rated thrust
6. Number of ignitions in vacuum
7. Life and reliability

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TABLE 3

ERECTION OF TEST STANDS REQUIRED FOR ENGINE DEVELOPMENT
(Mean Values for Central Engine Development)

1. One H_2 and one O_2 pump test stand
2. One gas-producer test stand
3. Two turbopump test stands
4. Two engine test stands
5. One Battleship test stand per rocket stage
6. Diverse handling test stands

TABLE 4

SOME DEVELOPMENT TASKS FOR ENGINE TEST STANDS

1. Development of the combustion chamber in terms of:
   a) Maximum characteristic velocity
   b) Control of heat transfer
   c) Dynamic stability of combustion
2. Development of the jet nozzle in terms of maximum thrust coefficients
3. Development of the ignition and start-up system in terms of
   a) Reliable ignition and re-ignition in vacuum
   b) As short as possible a start-up time
4. Development of coordination of combustion chamber, turbopumps, control and command systems
5. Development of Cardanic control
6. Acceptance test for proof of specifications fulfillment

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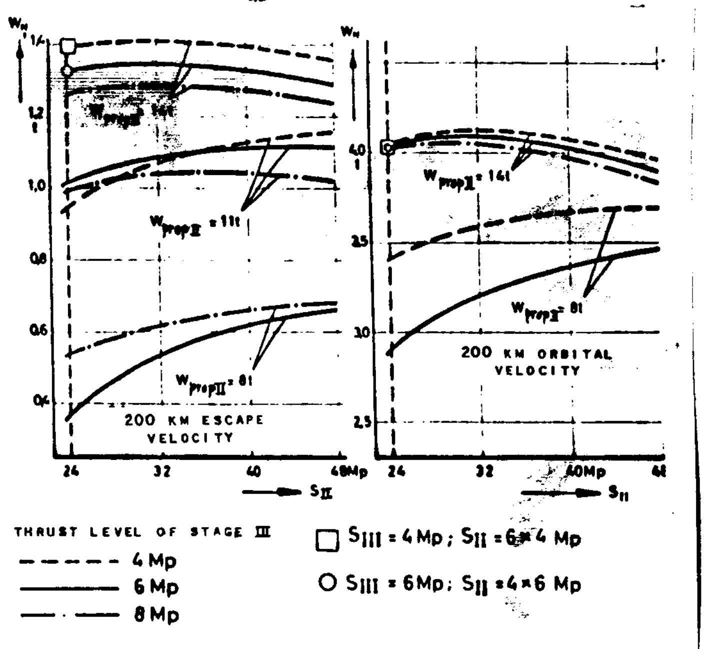

Fig.1 Payload W_N in the Preliminary Project of a Three-Stage ELDO-B Launch Vehicle, as a Function of the Thrust Level S_{II} of the Second Stage for Various Thrust Levels S_{III} in the Third Stage. (Liftoff mass W_A = 110 tons; S_I = = 144 Mp, upper-stage mass including payload, 23 tons)

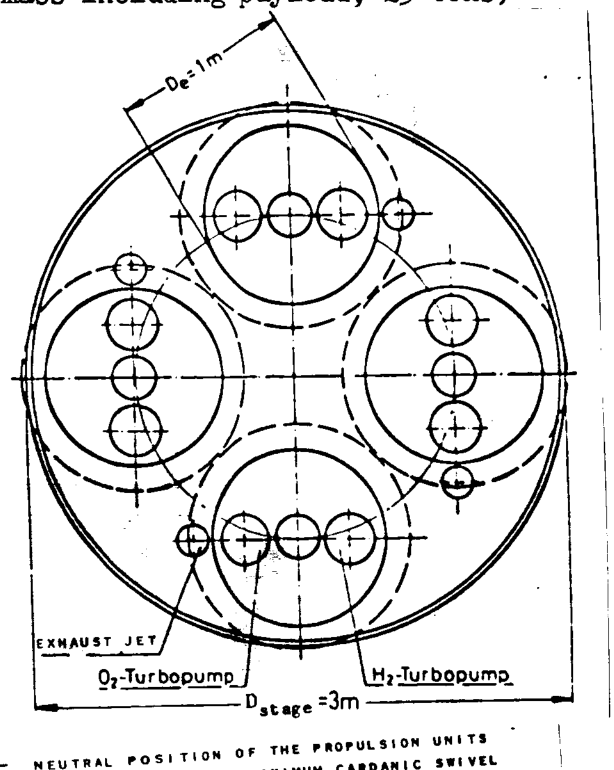

Fig.2 [illegible] of the Engine Layout in the Second Stage of the ELDO-B2 Launch Vehicle

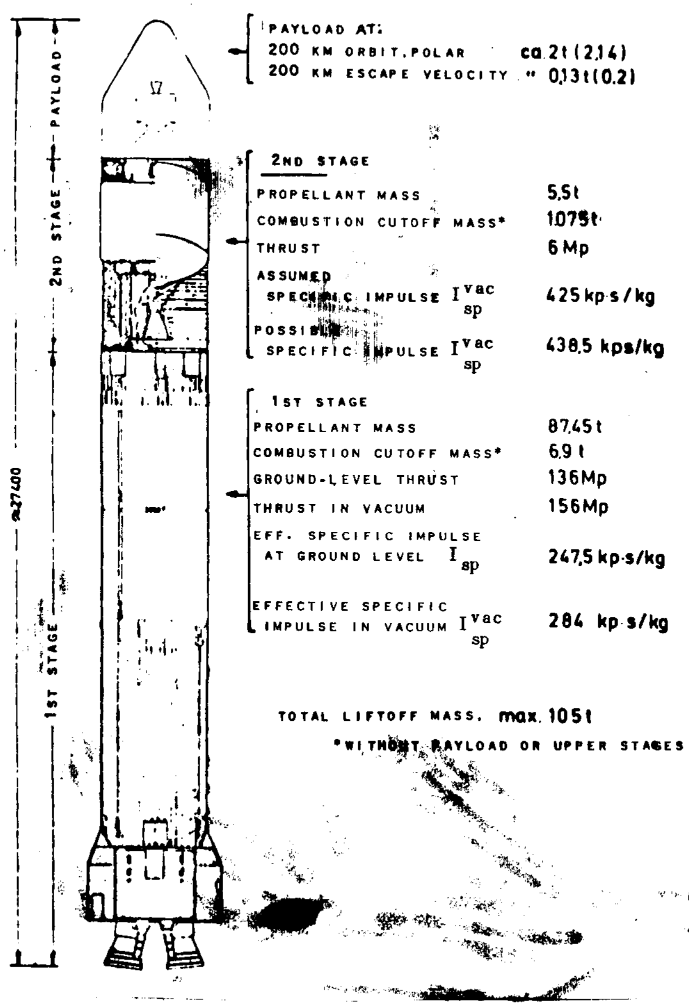

Fig.3 Project of a Two-Stage Launch Vehicle ELDO-B1

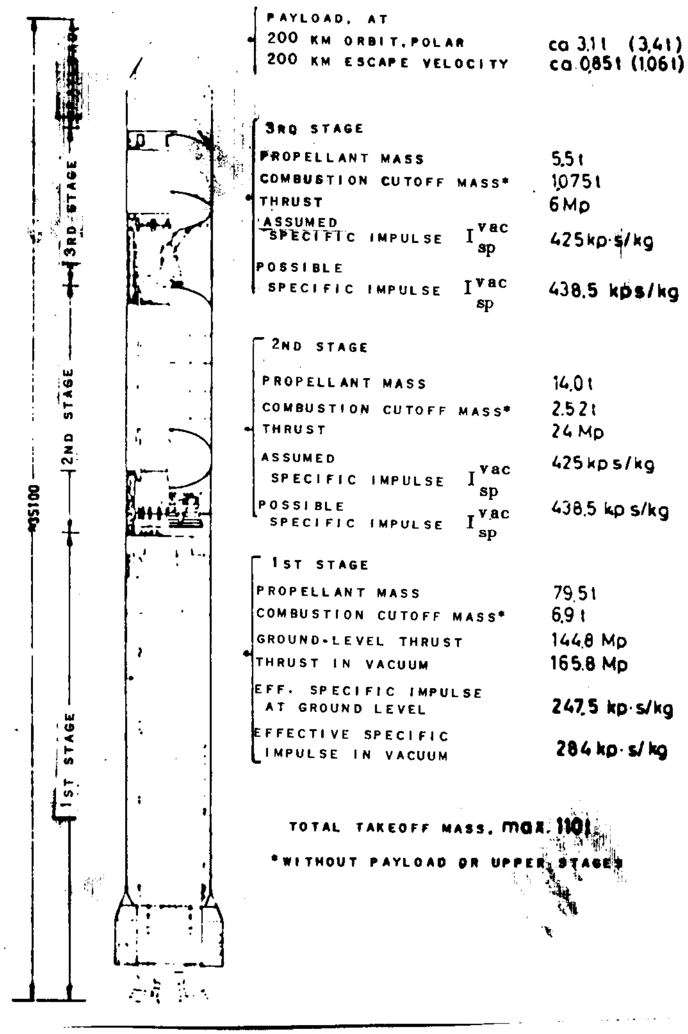

Fig.4 Project of a Three-Stage Launch Vehicle ELDO-B2

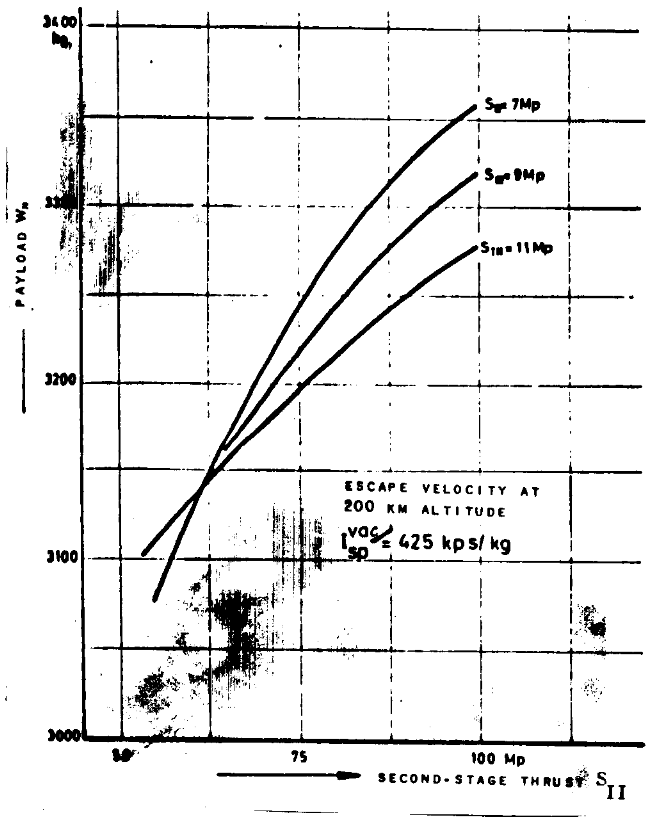

Fig.5 Payload W_N of a Three-Stage Launch Vehicle with a Liftoff Mass of 210 tons, as a Function of the Thrust Level S of the Second and Third Stage

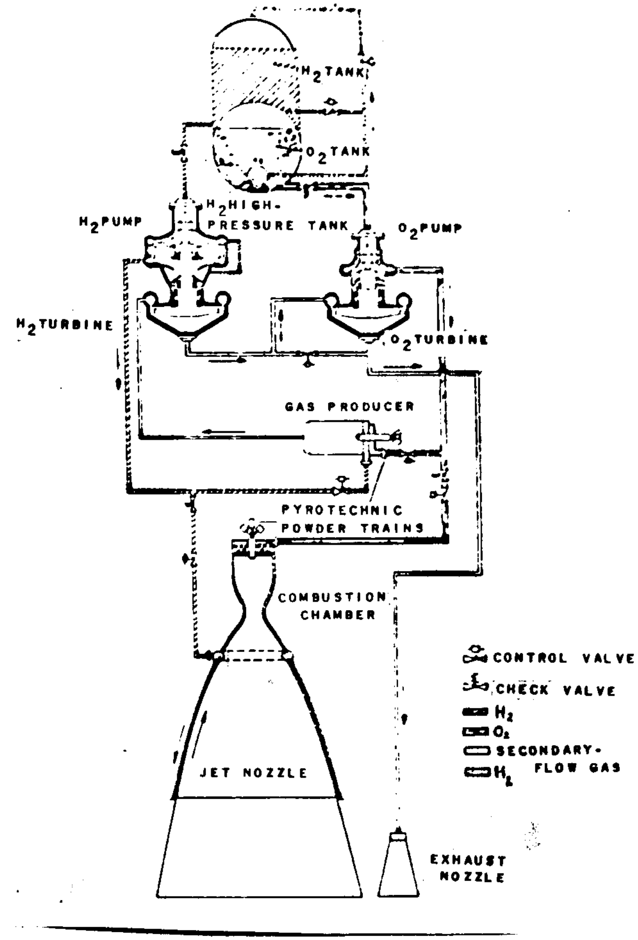

Fig.6 Schematic Presentation of the Engine System with Two Turbopumps

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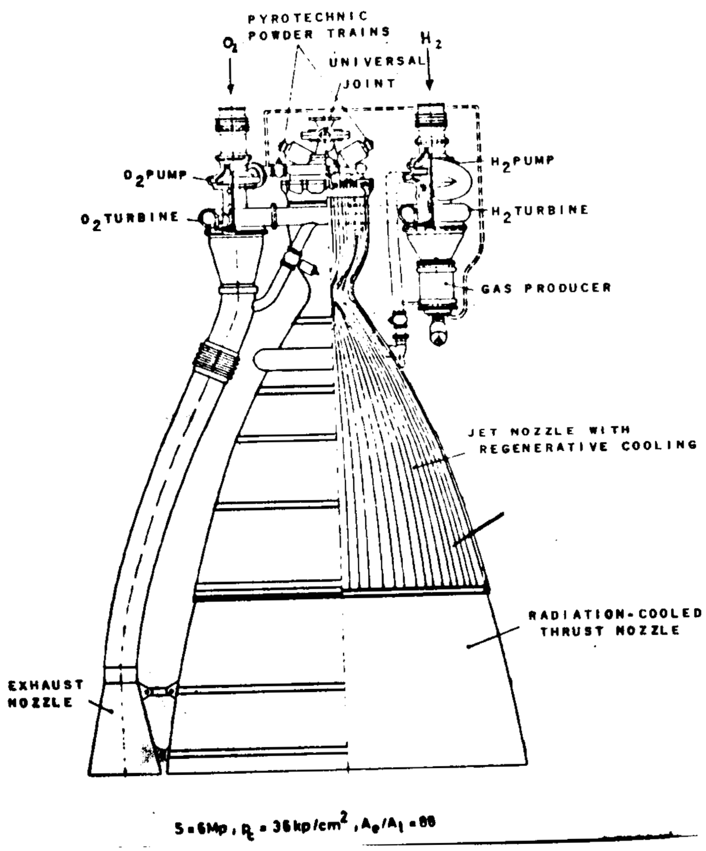

Fig.7 Schematic Presentation of the High-Energy Standard Propulsion Unit, in the Variant with Two Turbopumps

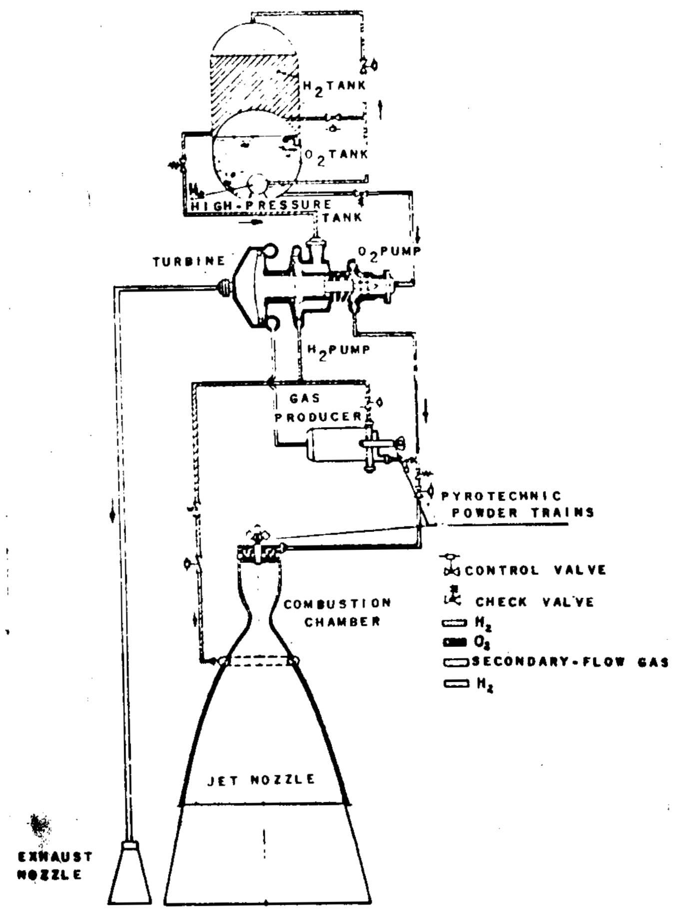

Fig.8 Schematic Presentation of the Engine System with Turbopump, in Single-Shaft Design

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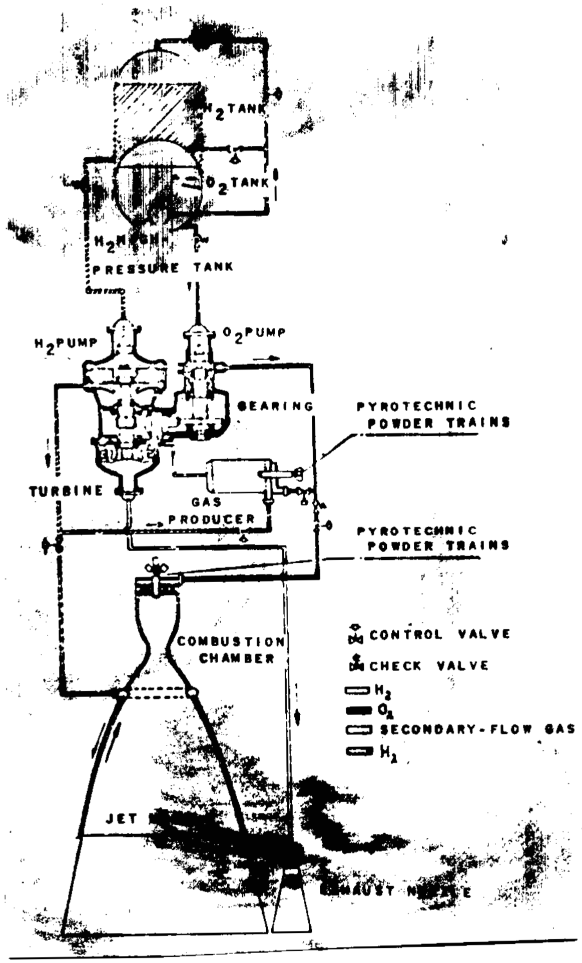

Fig.9 Schematic Presentation of the Engine System with Turbopump and Gearing

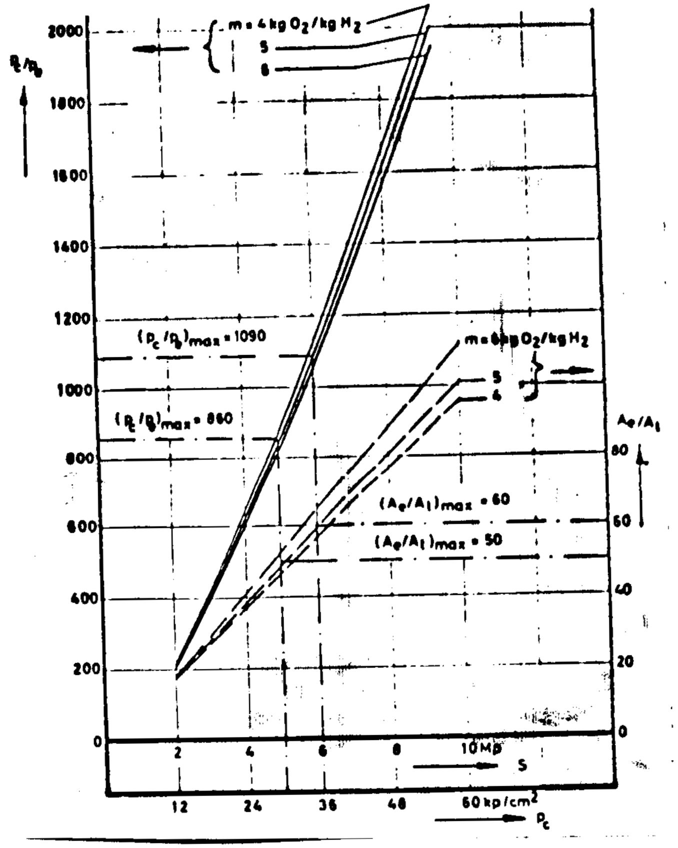

Fig.10 Maximum Area Ratio A_e/A_t and Expansion Ratio p_c/p_e of the Thrust Nozzle, as a Function of the Thrust S and the Combustion-Chamber Pressure p_c for Diffuser Compression of the Exhaust Gases to Ambient Pressure

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/18

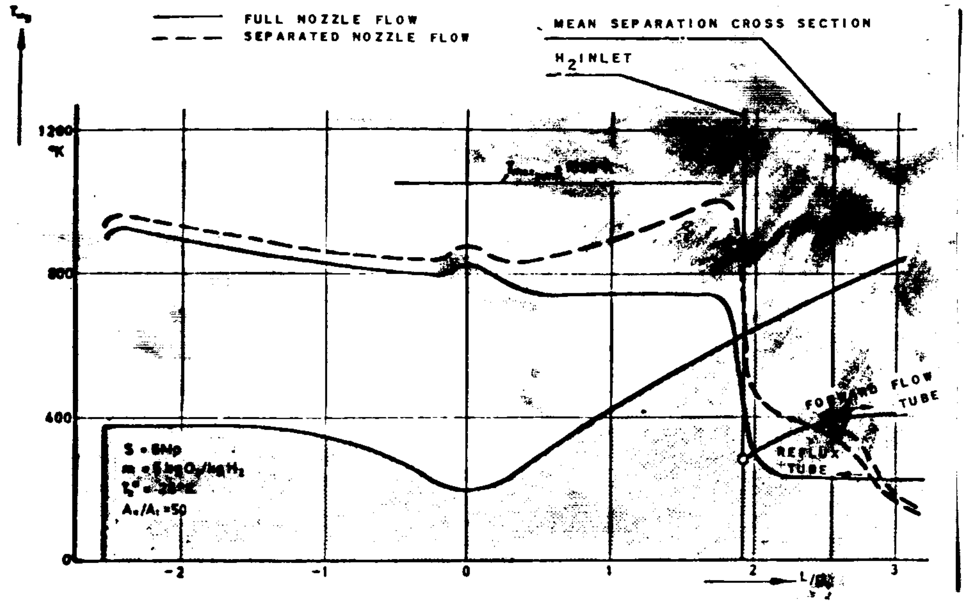

Fig.11 Wall Temperature T_{w_g} Plotted against Combustion-Chamber Length L/D_t at Full-Flow and Separated Nozzle Flow, under Atmospheric Counterpressure

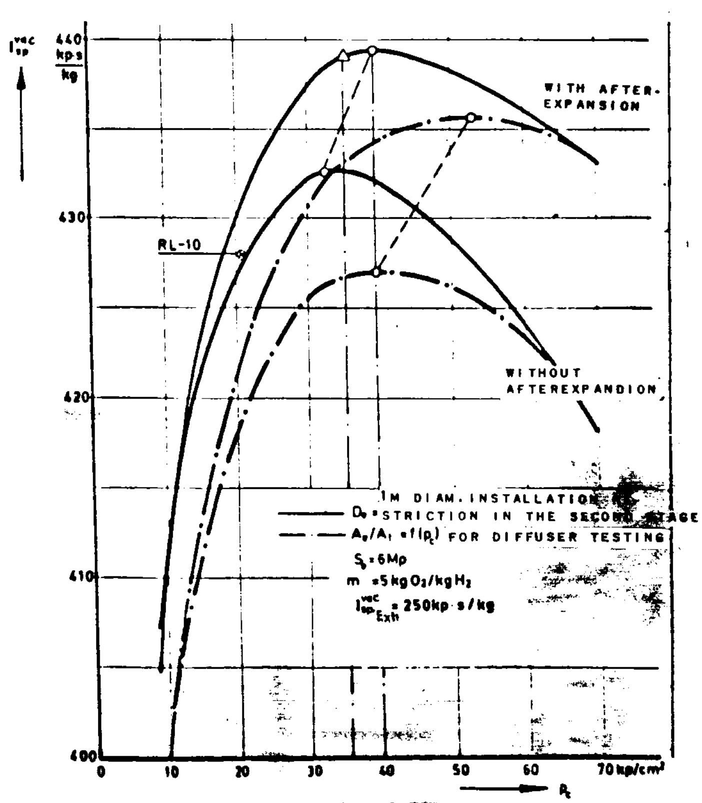

Fig.12 Effective Specific Impulse I_{sp}^{vac} of the Engine with and without After-expansion of the Turbine Exhaust Gases, as a Function of the Combustion-Chamber Pressure p_c

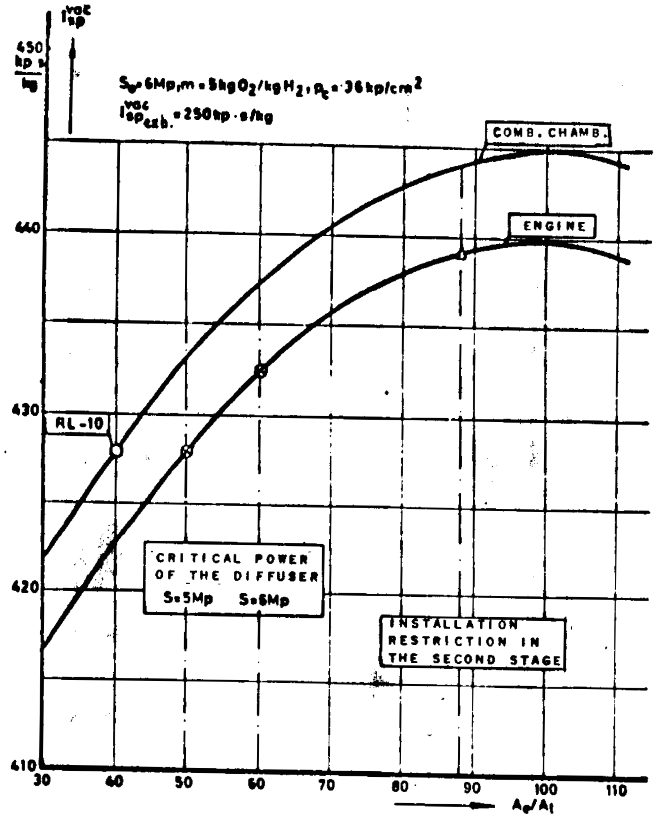

Fig.13 Effective Specific Impulse I_{sp}^{vac} of Combustion Chamber and Power Plant as a Function of the Jet Nozzle Area Ratio A_e/A_t

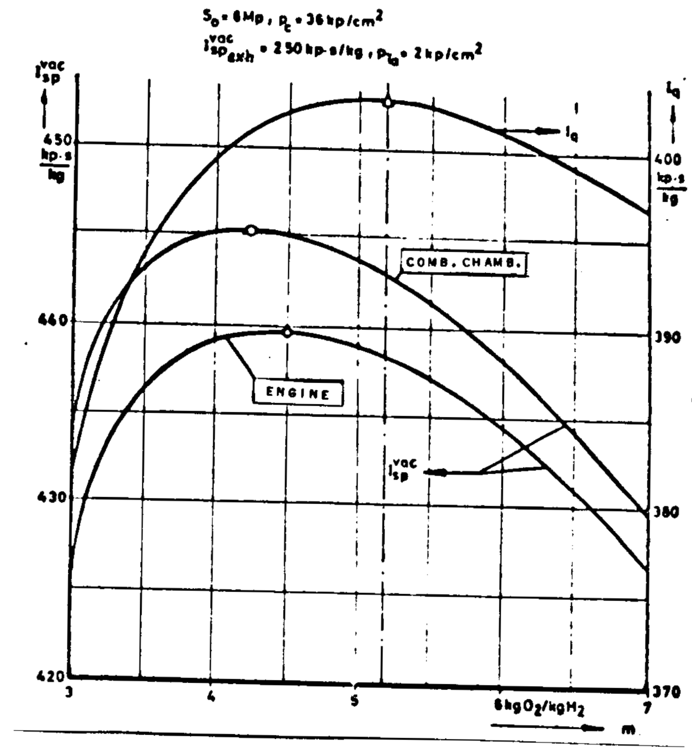

Fig.14 Maximum Effective Specific Vacuum Impulse I_{sp}^{vac} as a Function of the Mixture Ratio m in the Combustion Chamber, under Consideration of the Installation Restriction in the Second Stage, and System-Specific Impulse I_q of the Third-Stage Engine
(Weight of propulsive unit, pressure-gas system, and cylindrical tank section taken into consideration)

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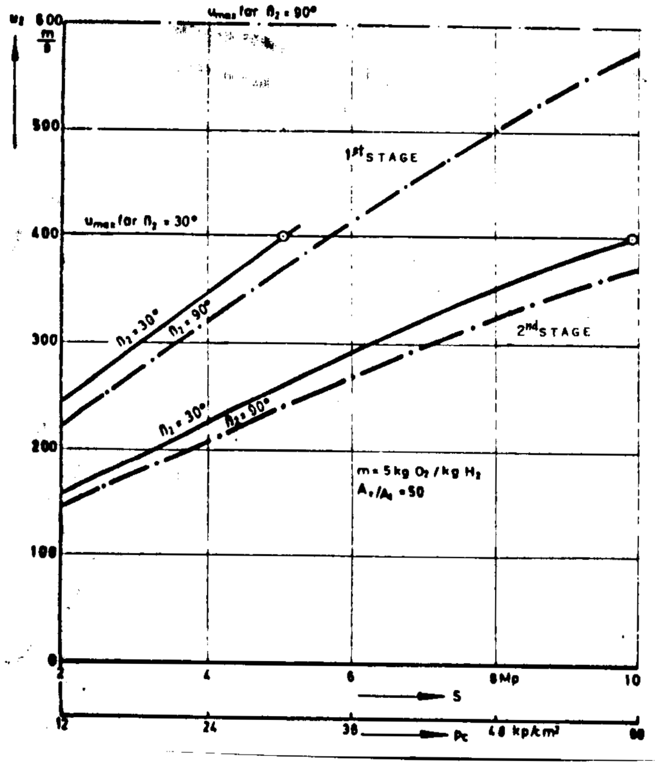

Fig.15 Peripheral Speeds u_2 of Hydrogen Pumps Plotted against the Thrust S, for Various Numbers of Stages and Blade Exit Angles β_2

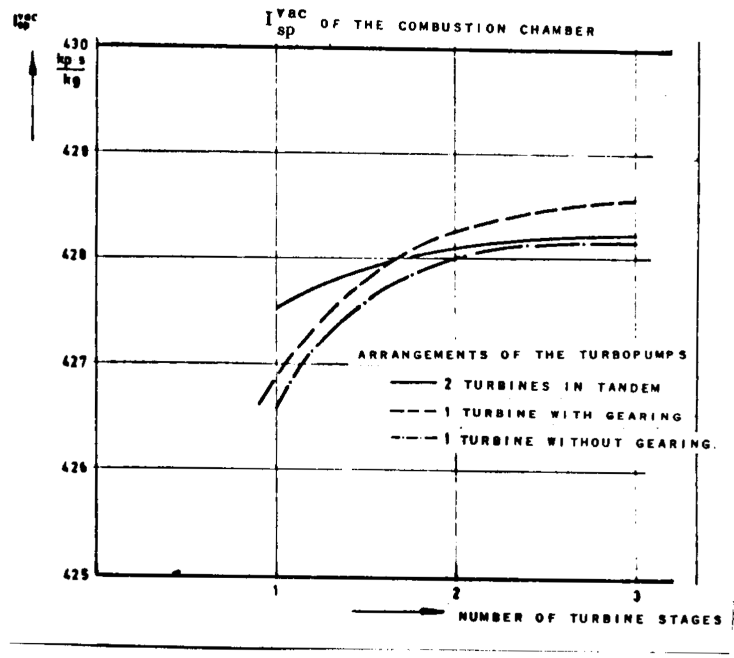

Fig.16 Effective Specific Vacuum Impulse I_{sp}^{vac} as a Function of the Number of Stages of the Turbine Drive

S = 6 Mp; m = 5; p_c = 36 kp/cm^2; I_{sp}^{vac} = = 430 kp/sec/kg (Assumption); m_{gas} = = 1.0 kg O_2/kg H_2; $I_{sp_{exh}}$ on expansion up to the condensation limit; $p_{exit\ turb}$ = 1 kp/cm^2

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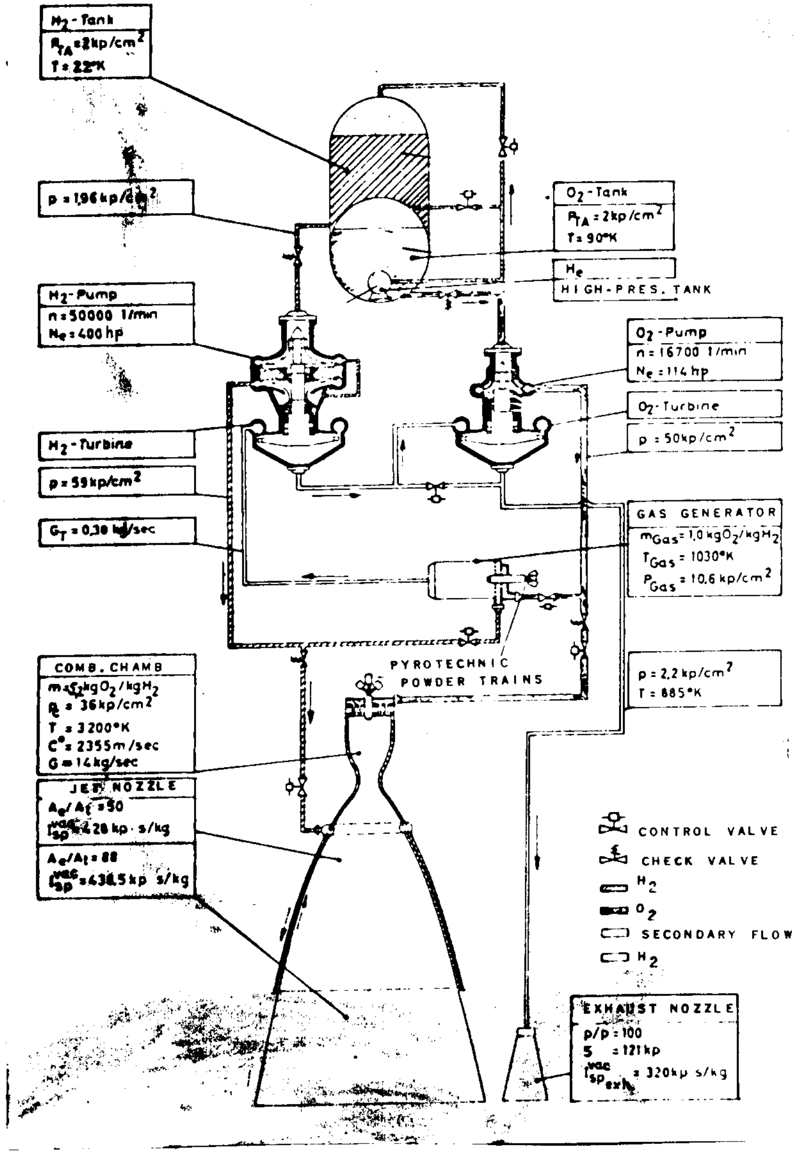

Fig.17 Example of Layout for the 6-Mp H_2/O_2 Engine with Propellant Feed by Two Turbopumps

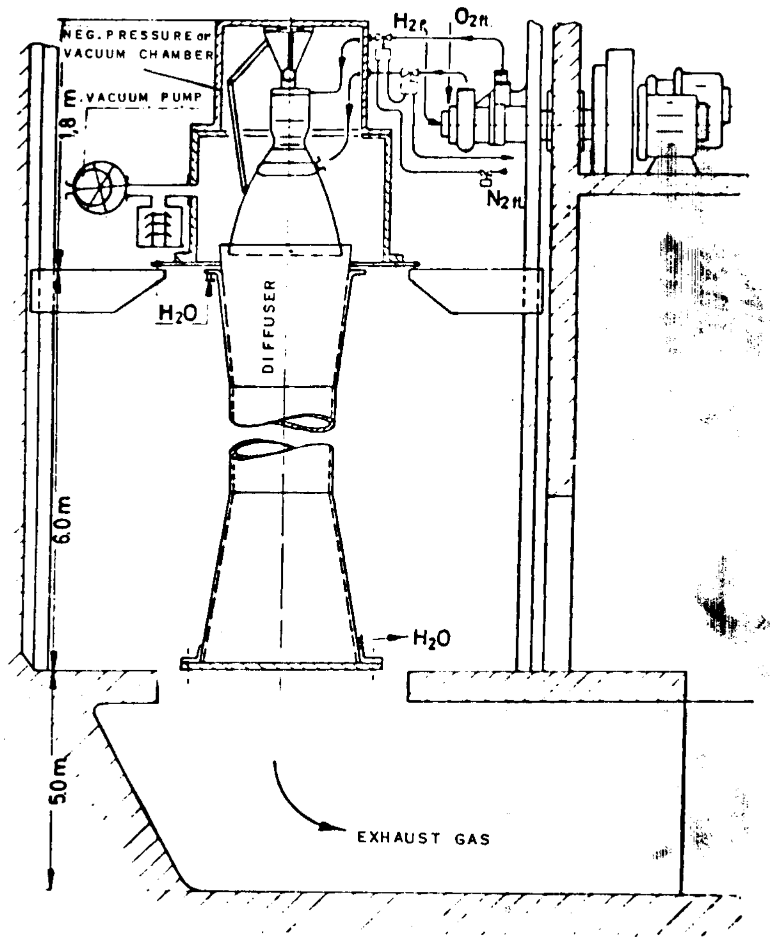

Fig.18 Example of a Vertical Engine Development Test Stand, in Schematic Presentation
Experimental setup for the development of combustion chamber and thrust nozzle at $(A_e/A_t)_{max}$ = = 50 - 60

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